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RESEARCH MEMORANDUM

EFFECT OF FUEL NOZZLE PROTRUSION ON TRANSIENT AND STEADY-STATE TURBOJET COMBUSTOR PERFORMANCE

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To ____________________

By authority of Nasa TPA 7 Effective Date 5-29-59
NB 7-6-59

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

February 23, 1955

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EFFECT OF FUEL NOZZLE PROTRUSION ON TRANSIENT AND STEADY-STATE TURBOJET COMBUSTOR PERFORMANCE

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SUMMARY

The effect of small variations in the axial position of the liner with respect to the nozzle on limiting rates of change of fuel flow (acceleration limits) and steady-state combustion efficiencies in a single tubular combustor was determined. Data were obtained with two liner configurations at three combustor-inlet conditions simulating 25,000, 40,000, and 50,000 feet altitude and a constant engine rotor speed of 70-percent rated at zero flight Mach number.

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Nozzle position had a marked influence on acceleration limits at all three altitude - rotor speed conditions. Within the range of relative nozzle positions that would be caused by thermal expansion of the liner, the acceleration limits varied through a four-fold range. The poorest acceleration characteristics were obtained with the nozzle protruding into the combustor liner. Steady-state combustion efficiencies were also affected by nozzle position, but in an opposite manner; the best efficiency performance was obtained with the nozzle protruded. These results serve to point out a combustor installation detail that affects combustion performance in combustor systems of the type used in this investigation.

INTRODUCTION

Among the problems associated with turbojet engine operation at high altitude is the inability of the engine to accelerate rapidly in response to increased fuel flows. Research is being conducted at the NACA Lewis laboratory to determine the factors affecting engine acceleration. As part of this research, an investigation of the effect of axial location of the fuel injector, relative to the liner, on the combustion process during fuel acceleration in a single tubular combustor is reported herein.

Results of an investigation reported in reference 1 described combustion response to rapid fuel-flow changes in a tubular combustor at two simulated altitude - rotor speed conditions. Limiting rates of change of

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fuel flow (acceleration limits) were determined and the effects of certain air-flow variables on the transient combustion characteristics were studied. Further studies in the same combustor indicated that variations in combustor component alinement in day-to-day testing varied the acceleration limits obtained. Investigation showed that the axial position of the fuel nozzle, relative to the combustor liner, was a possible cause of variation in results. The relative position of the nozzle is a function of the care taken in installation and of the thermal expansion of the liner. With the particular tubular combustor used, the combustor liner was anchored at its downstream end, and thermal expansion of the 3533
liner, the outer housing of the combustor, and the connecting setup ducting resulted in the liner shifting upstream. The fuel nozzle was rigidly mounted on the inlet diffuser; the shifting of the liner thus caused a variation in the axial location of the fuel nozzle relative to the liner. Accordingly, an investigation of the effects of nozzle location or protrusion into the primary combustion zone on fuel acceleration limits and steady-state performance was conducted.

In order to provide fixed positioning of the nozzle with respect to the liner during individual series of tests, the liner was attached to the fuel nozzle assembly, and thermal expansion of the liner occurred in a downstream direction. The nozzle was a standard dual-entry duplex fuel nozzle. Data were obtained at three nozzle protrusion positions with two slightly different liner primary zone configurations. Three combustor-inlet air conditions were investigated corresponding to 70 percent rated rotor speed at 25,000, 40,000, and 50,000 feet altitude in a reference turbojet engine. The data are analyzed to indicate the effect of fuel-nozzle-protrusion position on the fuel acceleration limits and combustion efficiencies for a range of fuel-air ratios at the simulated engine operating conditions chosen. Descriptions of the special apparatus and instrumentation used are presented.

APPARATUS AND INSTRUMENTATION

A single combustor from a J35-C-3 turbojet engine was used in this investigation. The combustor was connected to the laboratory air supply as shown diagrammatically in figure 1. The air-flow rate and pressure were regulated by remote-control valves upstream and downstream of the combustor. Air flow was measured by means of a variable-area orifice. In order to assure a uniform air and exhaust supply free of line surges, choke plates were placed in the inlet and exhaust ducting of the combustor. Location and construction of these choke plates are shown in figure 2. The inlet choke plate admitted air through fifty 1/4-inch-diameter holes. The outlet choke-plate assembly consisted of two slotted plates, one of which was movable with respect to the other, permitting a range of flow areas to be selected. The inlet choke plate and the outlet choke assembly were installed in the ducting at positions corresponding to the last stage of the compressor and the turbine nozzle diaphragm in the full-scale engine, respectively.

Two combustor liner configurations were used: (1) the production configuration, and (2) the production configuration modified by removing the spark-plug-hole cover plate, which allowed an additional 1/2 square inch of area for air entry into the primary zone.

Combustor Fuel System

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Two fuel systems were used to obtain the required flow rates for the steady-state and the transient phases of the investigation. A conventional fuel system containing fuel storage drums, pumps, measuring rotameters, connecting piping, and manual regulating valves was used to obtain steady-state combustion data. A separate fuel system containing a pressurized container, motorized flow control valve, and surge chambers was employed to obtain transient combustion data. A more detailed description of the fuel acceleration system is given in reference 1. The fuel used was MIL-F-5624A, grade JP-4 (NACA 52-288). A standard dual-entry duplex nozzle was altered to permit fixed positioning of the liner to the nozzle. A diagrammatic sketch of the fuel nozzle and combustor showing the method of attaching the liner to the nozzle is presented in figure 3. Nine channels were cut into the nozzle body and the dome was attached to the desired channel by three set screws inserted through the nozzle dome ring. This arrangement permitted the tip of the nozzle to be positioned at the face of nozzle ring and up to 5/8 of an inch downstream. With the unaltered combustor system the axial position of the nozzle may vary by approximately 1/4 to 5/16 of an inch due to liner thermal expansion. This estimated variation was checked by observing the change in nozzle position while operating the combustor with the unaltered liner and nozzle arrangement over the range of inlet conditions used in the investigation. Three protrusion positions of 0, 5/16, and 5/8 inch were chosen to more than cover the possible range of nozzle protrusion distances, whether caused by liner expansion or by assembly errors.

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Instrumentation

Combustor-inlet air temperature was measured by two single-junction iron-constantan thermocouples located at station 1 (fig. 1). Steady-state combustor-inlet static pressure was measured by means of static taps located at station 2 (fig. 1). Transient combustor-inlet static pressure was measured at the same station (2) with a diaphragm-type differential pressure pickup and was recorded on an oscillograph.

The combustor-outlet temperature was measured by three five-junction chromel-alumel thermocouple rakes located at station 3 (fig. 1). These thermocouples were connected through an averaging circuit to a potentiometer and were used to indicate steady-state outlet temperatures and temperatures before and after fuel accelerations. The rapid variations in combustor-outlet temperature during the acceleration process were indicated by a single thermocouple that was compensated for thermal lag. The single thermocouple, located between the rakes at station 3, consisted of 0.010-inch diameter wires butt-welded between two heavier support wires. The position of the single thermocouple junction in the gas stream was selected to indicate the same temperature as the average reading of the 15 outlet thermocouples during steady-state operation. The temperature indications were recorded by an oscillograph. A detailed discussion of the methods of thermocouple compensation is given in reference 1. The theory of compensation is presented in reference 2. 3533

The transient fuel-flow rate was measured with a pressure differential pickup and a constant-current hot-wire anemometer. The pressure differential pickup was connected across an orifice in the large-slot nozzle supply line. This orifice served to divert fuel to the small slots, providing improvement in spray formation at low fuel flows. The pressure pickup, properly calibrated, measured steady-state fuel flow accurately and was used to indicate the flow before and during acceleration. The anemometer had a higher frequency response but was less accurate; the anemometer was used to determine the time elapsed during the fuel-flow change. The signals obtained from both flow measuring devices were recorded on an oscillograph.

PROCEDURE

Test Conditions

Combustor transient response characteristics and steady-state temperature rise were studied at the following operating conditions:

Simulated flight conditions		Inlet static pressure, in. Hg abs	Inlet-air temperature, °F	Inlet-air flow, lb/sec	Reference velocity, ft/sec	Outlet temperature before acceleration, °F
Altitude, ft	Rated rotor speed, percent					
50,000	70	9.3	80	0.9	82	675-700
40,000	70	15.2	80	1.43	80	675-700
25,000	70	28	80	2.7	82	290

The two highest altitude conditions simulated operation of the combustor in a 4.7-pressure-ratio turbojet engine at a flight Mach number of 0, with the exception of inlet-air temperature. The 25,000 feet altitude condition simulated operation in the same engine at the same speed; however, the outlet temperature was reduced from the required value (700° F) to 290° F. The outlet temperature was set at this lower value to obtain limiting fuel acceleration rates of the same order of magnitude as those obtained at the two higher altitude conditions. No acceleration limits could be obtained within the range of fuel acceleration rates supplied by the equipment at 25,000 feet altitude when the outlet temperature was 700° F. The reference velocity values quoted are based on the maximum cross-sectional area of the combustor (0.48 sq ft) and the inlet-air density.

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Test Procedure

Combustor steady-state temperature-rise data were obtained at each of the operating conditions noted above. At each test condition data were recorded over a wide range of fuel-air ratios.

Transient combustor response data were obtained in the following manner. The transient instrumentation was first calibrated against the steady-state instrumentation. The acceleration fuel system was then adjusted and energized to provide the desired rate of increase in fuel-flow rate. For selected values of final fuel flow, the slope of the fuel acceleration was increased by readjusting components of the accelerating system until combustion blow-out occurred or the limit of the fuel system was reached. This procedure was repeated for each combustor-inlet condition with three fuel nozzle protrusion positions and two liner configurations.

Method of Analysis

The fuel acceleration rates referred to herein represent the fuel slopes and were computed as the change of fuel-air ratio per unit of time. Figure 4 shows a sketch of a typical fuel ramp trace as recorded by the pressure differential pickup. The acceleration rate was calculated by subtracting the initial fuel-air ratio from the final fuel-air ratio and dividing the difference by the amount of time (sec) between the point on the trace where the acceleration begins and the point where the fuel flow first reaches the desired final fuel flow. A fuel "overshoot" was indicated during rapid fuel accelerations as shown in figure 4. This overshoot could not be eliminated with the fuel systems used to obtain these accelerations.

RESULTS

Transient Combustion Performance

Analysis of the oscillograph traces showing the variation of fuel flow, inlet-static pressure, and outlet temperature indicated that the combustion process during acceleration followed one of three alternate paths:

(1) The additional fuel may ignite and burn stably, resulting in increased temperature rise.

(2) The additional fuel may ignite, burn temporarily at a higher temperature level, and then blow out.

(3) The combustion may blow out immediately after fuel acceleration 3533
is begun.

The second and third types of combustion response represent, of course, unsuccessful attempts to accelerate. For the successful acceleration data the outlet temperature and static pressure first decreased with the introduction of additional fuel and then increased as the fuel burned. This dip and rise sequence was more pronounced as the simulated altitude and fuel acceleration rate increased. These characteristics of transient combustion response are similar to those reported previously (ref. 1) and were not altered by nozzle protrusion position.

The transient combustion performance data obtained with the three nozzle protrusion positions at the three simulated altitude - rotor speed conditions are presented in tables I, II, and III. Acceleration rate is plotted against the final fuel-air ratio in figure 5; these results were obtained with the unmodified liner. Different symbols are used to identify the different nozzle protrusion distances, with unsuccessful acceleration denoted by tailed symbols. Curves are interpolated through the data to represent limits of successful acceleration. The rich-blow-out fuel-air ratios (steady-state) were approached only at the 50,000 feet altitude condition; the rich-blow-out region is included on figure 5(c). The unsuccessful acceleration data reported for the 40,000 and 25,000 feet altitude conditions were all "quench-out" points (the third combustion response path described previously). With the rapid fuel accelerations necessary to establish these quench-out points, the transient fuel-flow rates always "overshot" the final fuel flow (see fig. 4). This fuel flow overshoot could not be eliminated by modifications to the surge chambers in the fuel line; the effects of this overshoot on the acceleration limits were not determined.

The nozzle protrusion position had a marked influence on the acceleration limits obtained at all three simulated altitude - rotor speed conditions. In most cases acceleration limits decreased with depth of protrusion. The fastest rates of successful fuel acceleration were obtained when the nozzle protrusion was zero; because of equipment limitations the acceleration limits for zero nozzle protrusion could be determined only at the 40,000 feet altitude condition. At 50,000 feet altitude unsuccessful accelerations were obtained when the final fuel-air ratios were large enough to cause steady-state rich blow-out, but no limits were obtained at lower fuel-air ratios.

The data obtained from tests with the modified liner (spark-plug-hole cover plate removed) are shown in figure 6; the data are again plotted as fuel acceleration rate against fuel-air ratio after acceleration. The dotted lines represent the limits obtained with the unmodified liner. The limits obtained with the modified liner are similar to those obtained with the unmodified liner, except those obtained at 40,000 feet altitude with the zero protrusion position. The trend of increasing limits with decreasing nozzle protrusion distance is again apparent.

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Steady-State Combustion Performance

The steady-state combustion performance data obtained at conditions simulating a constant engine rotational speed and three different altitudes are presented in table IV. A comparison of outlet temperatures against fuel-air ratios for the different fuel-nozzle protrusion distances investigated is shown in figure 7. Included in figure 7 are lines of constant combustion efficiency; by interpolating between these lines the combustion efficiency value of each data point can be approximated. These lines of constant combustion efficiency were determined from the charts of reference 3, and were computed as the ratio of enthalpy rise across the combustor to the heating value of the fuel. The tailed symbols represent data obtained with the modified liner having the spark plug hole open, allowing additional air into the primary zone.

The steady-state combustion data show the expected trend of decreasing combustion efficiency with increased simulated altitude. The differences in performance obtained with the three nozzle positions also increased with increased altitude. The combustion efficiency spread among the data obtained with the three nozzle positions is approximately 10 percent at 50,000 feet and 5 percent at 25,000 feet altitude, excluding the lowest fuel-air ratio points. For the conditions investigated, the best temperature-rise performance (and combustion efficiency) was obtained with the nozzle protruding either 5/16 or 5/8 inch into the combustor. Also, little or no difference in combustion efficiency was obtained with the production liner and the modified liner.

DISCUSSION

The investigation of the effect of nozzle protrusion depth on transient combustion characteristics showed that protrusion depth has a marked influence on the ability of the combustor to burn additional fuel injected in a short time. Acceleration rate limits are plotted against nozzle position in figure 8 for the three altitude - rotor speed conditions. These limits were taken from figure 5 for a final fuel-air ratio of 0.026.

The range of nozzle positions resulting from thermal expansion of the liner in a full-scale engine operating at various altitude - rotor speed conditions is included in figure 8. For this range of nozzle positions, the limits vary from 0.017 to 0.076 fuel-air ratio per second at 40,000 feet altitude, indicating a wide variation in the ability of the combustion process to successfully respond to added fuel. For the complete range of nozzle positions and combustor-inlet conditions covered in the investigation, the data show differences in acceleration limits of approximately one order of magnitude. The importance of careful alinement of combustor components in the engine is emphasized by these results. It should be realized that combustion response to fuel acceleration rates in the speed range near the acceleration limit curves was not always the same. However, this irreproducibility was not of sufficient magnitude to account for the wide variation in limits obtained with different nozzle positions.

The results shown in figure 8 provide a possible explanation for the scattering of blow-out data obtained with full-scale turbojet engines during acceleration tests of reference 4 conducted in the NACA Lewis altitude wind tunnel. The wind tunnel investigation was conducted with two models of the J47D engine that were aerodynamically similar. The rates of fuel acceleration used were in approximately the same range as those used in the single combustor tests. Two tubular combustor configurations that provided slightly different primary-air patterns were used in these engines. These combustion chambers were of the same general tubular type, were equipped with the same type of dual-entry duplex fuel nozzles, and were assembled into the outer housing in a manner that would result in changes in nozzle protrusion with changes in thermal expansion of the liner.

No appreciable difference was noted in acceleration performance between the two different liner configurations in the engine tests of reference 4. This result agrees with that obtained in the single combustor tests; it was found that tolerance to fuel acceleration was not greatly influenced by small changes in air-hole area in the primary zone of the combustor.

The steady-state combustion efficiencies were affected by nozzle position in the single combustor tests. The best efficiency performance was obtained with the nozzle protruding into the combustion chamber, a condition which provided the worst acceleration performance. These opposed effects of nozzle position point up the fact that combustor design must often be compromised to obtain the best over-all performance characteristics. If fuel-air ratio changes of 0.01 per second are considered rapid enough for engine accelerations, then designs providing maximum combustion efficiency performance would be preferred. Obviously, these results are restricted to combustion systems of the type investigated, as the optimum location of the nozzle for this combustor might not coincide with the optimum location for other types of combustor and fuel nozzle systems.

The variance in steady-state performance caused by nozzle location shows that means should be provided on single combustor test rigs to duplicate the exact nozzle position relative to the liner such as would occur in a full-scale engine under identical operating conditions.

SUMMARY OF RESULTS

An investigation was conducted to determine the effect of fuel nozzle axial location on fuel acceleration limits and steady-state combustion efficiencies of a single tubular combustor. The highest acceleration limits were obtained when the end of the nozzle was nearly flush with the contour of the dome inner wall. Within the range of nozzle protrusion predicted due to liner expansion, the acceleration limits varied through a four-fold range.

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Nozzle locations also had an effect on the combustion efficiencies obtained and, in general, the highest efficiencies were obtained with the maximum protrusion depth of the nozzle investigated. It should be realized that these results apply only to combustion systems of the type investigated. They serve to point out a combustor installation detail that markedly affects combustion performance.

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Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, November 8, 1954

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3. Turner, L. Richard, and Bogart, Donald: Constant-Pressure Combustion Charts Including Effects of Diluent Addition. NACA Rep. 937, 1949. (Supersedes NACA TN's 1086 and 1655.)

4. Conrad, E. William, Bloomer, Harry E., and Sobolewski, Adam E.: Altitude Operational Characteristics of a Prototype Model of the J47D (RXI-1 and RXI-3) Turbojet Engines with Integrated Electronic Control. NACA RM E51E08, 1952.

TABLE I. - TRANSIENT COMBUSTION PERFORMANCE DATA FOR 25,000 FEET

SIMULATED ALTITUDE

[Simulated rotor speed, 70-percent rated; inlet static pressure, 28.0 in. Hg abs; air flow, 2.7 lb/sec; inlet temperature, 80° F; reference velocity, 82 ft/sec; approx. initial fuel-air ratio, 0.004; initial outlet temperature, 290° F.]

(a) Production liner (unmodified) 3533

Run	Final fuel-air ratio	Time for acceleration, sec	Acceleration rate, fuel-air-ratio change per second	Combustor response	Nozzle protrusion position, in.
124	0.0182	0.14	0.10	Successful	0
125	.0232	.16	.12	Unsuccessful	
182	.0187	.54	.028	Successful	5/16
183	.0187	.40	.038	Unsuccessful	
184	.0157	.36	.034	Unsuccessful	
185	.0150	.44	.026	Unsuccessful	
186	.0228	.14	.014	Successful	
187	.0228	.84	.023	Unsuccessful	
188	.0144	.75	.014	Successful	
189	.0144	.54	.020	Unsuccessful	
150	.0179	.56	.026	Successful	5/8
151	.0179	.24	.061	Unsuccessful	
152	..0179	.38	.039	Unsuccessful	
153	.0187	1.0	.015	Successful	
154	.0187	.66	.023	Unsuccessful	
155	.0212	.76	.024	Successful	
156	.0212	.52	.034	Unsuccessful	

(b) Modified liner

Run	Final fuel-air ratio	Time for acceleration, sec	Acceleration rate, fuel-air-ratio change per second	Combustor response	Nozzle protrusion position, in.
86	0.0206	0.18	0.098	Successful	0
25	.0126	.65	.013	Successful	5/16
26	.0126	.5	.017	Unsuccessful	
27	.0145	.68	.016	Successful	
28	.0145	.49	.021	Unsuccessful	
29	.0157	.87	.014	Successful	
30	.0157	.58	.020	Unsuccessful	
38	.0128	.80	.012	Successful	5/8
39	.0128	.53	.018	Unsuccessful	
40	.014	.88	.012	Successful	
41	.014	.67	.016	Unsuccessful	
42	.0154	.65	.018	Successful	
43	.0154	.54	.022	Unsuccessful	

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TABLE II. - TRANSIENT COMBUSTION PERFORMANCE DATA FOR 40,000 FEET

SIMULATED ALTITUDE

[Simulated rotor speed, 70-percent rated; inlet static pressure, 15.2 in. Hg abs; air flow, 1.43 lb/sec; inlet temperature, 80° F; reference velocity, 80 ft/sec; approx. initial fuel-air ratio, 0.009; initial outlet temperature, 675°-700° F.]

(a) Production liner (unmodified)

Run	Final fuel-air ratio	Time for acceleration, sec	Acceleration rate, fuel-air-ratio change per second	Combustor response	Nozzle protrusion position, in.
126	0.0324	0.20	0.12	Successful	0
127	.0334	.22	.11	Unsuccessful	
128	.0363	.55	.050	Successful	
129	.0363	.40	.068	Unsuccessful	
130	.0377	.49	.058	Successful	
131	.0377	.36	.079	Unsuccessful	
139	.026	.12	.14	Successful	
140	.0288	.15	.13	Unsuccessful	
175	.0346	1.6	.016	Successful	5/16
176	.0346	.70	.037	Unsuccessful	
177	.0313	.94	.024	Successful	
178	.0313	.63	.035	Unsuccessful	
179	.0268	.62	.029	Unsuccessful	
180	.0241	.61	.025	Unsuccessful	
181	.0245	.90	.017	Successful	
163	.0291	1.3	.015	Successful	5/8
164	.0291	.74	.028	Unsuccessful	
165	.0278	.52	.037	Unsuccessful	
166	.0262	1.1	.016	Unsuccessful	
167	.0222	1.1	.012	Unsuccessful	
168	.0214	1.0	.013	Successful	

(b) Modified liner

Run	Final fuel-air ratio	Time for acceleration, sec	Acceleration rate, fuel-air-ratio change per second	Combustor response	Nozzle protrusion position, in.
71	0.0247	0.2	0.078	Successful	0
72	.0247	.17	.092	Unsuccessful	
73	.0281	.42	.045	Successful	
74	.0281	.32	.06	Unsuccessful	
75	.0315	.62	.036	Successful	
76	.0315	.46	.049	Unsuccessful	
14	.0208	.54	.022	Successful	5/16
15	.0208	.52	.023	Unsuccessful	
16	.0245	.7	.022	Successful	
17	.0245	.68	.024	Unsuccessful	
18	.0268	1.08	.016	Successful	
19	.0268	.78	.023	Unsuccessful	
49	.0284	1.85	.011	Successful	5/8
50	.0284	.86	.023	Unsuccessful	
51	.0284	1.24	.016	Unsuccessful	
52	.0252	1.26	.013	Successful	
53	.0252	.82	.020	Unsuccessful	
54	.0214	.78	.016	Unsuccessful	

TABLE III. - TRANSIENT COMBUSTION PERFORMANCE DATA FOR 50,000 FEET

SIMULATED ALTITUDE

[Simulated rotor speed, 70-percent rated; inlet static pressure, 9.3 in. Hg abs; air flow, 0.9 lb/sec; inlet temperature, 80° F; reference velocity, 82 ft/sec; approx. initial fuel-air ratio, 0.0115; initial outlet temperature, 675°-700° F.]

(a) Production liner (unmodified)

Run	Final fuel-air ratio	Time for acceleration, sec	Acceleration rate, fuel-air-ratio change per second	Combustor response	Nozzle protrusion position, in.
112	0.0262	0.22	0.070	Successful	0
113	.0307	.24	.082	Successful	
132	.026	.18	.15	Successful	
136	.0315	.20	.099	Unsuccessful	
137	.0315	.33	.060	Successful	
138	.0321	.70	.029	Unsuccessful	
193	.0306	1.0	.020	Successful	5/16
194	.0306	.60	.033	Unsuccessful	
195	.0268	.42	.038	Successful	
196	.0268	.36	.044	Unsuccessful	
197	.0231	.24	.051	Successful	
108	.0267	.40	.041	Unsuccessful	5/8
109	.0262	.60	.026	Unsuccessful	
110	.0262	.70	.023	Successful	
111	.0262	.60	.025	Unsuccessful	
157	.0304	.80	.025	Unsuccessful	
158	.0308	2.5	.0082	Unsuccessful	
159	.0278	.80	.022	Unsuccessful	
160	.0274	1.6	.011	Successful	
161	.0234	.65	.020	Successful	
162	.0234	.55	.026	Unsuccessful	

(b) Modified liner

Run	Final fuel-air ratio	Time for acceleration, sec	Acceleration rate, fuel-air-ratio change per second	Combustor response	Nozzle protrusion position, in.
88	0.0320	0.40	0.051	Unsuccessful	0
89	.0312	.55	.036	Successful	
90	.0330	3.4	.0064	Unsuccessful	
91	.0330	8.0	.0027	Unsuccessful	
92	.0312	6.5	.0031	Unsuccessful	
93	.0304	6.0	.0032	Unsuccessful	
94	.0278	7.0	.0024	Successful	
95	.0302	1.6	.0118	Unsuccessful	
96	.0294	1.8	.0101	Successful	
97	.0299	8.0	.0023	Successful	
98	.0293	.22	.082	Successful	
99	.0304	.26	.073	Unsuccessful	
100	.0307	1.0	.02	Unsuccessful	
101	.0301	4.5	.0043	Successful	
102	.0315	6.6	.0031	Successful	
103	.0327	5.6	.0039	Unsuccessful	
104	.0308	1.8	.011	Successful	
105	.0315	2.1	.0099	Unsuccessful	
106	.0301	.85	.023	Unsuccessful	
107	.0292	.88	.021	Successful	
108	.0301	.30	.10	Successful	
1	.0244	.31	.042	Successful	5/16
2	.0244	.30	.044	Unsuccessful	
3	.0272	.52	.031	Successful	
4	.0272	.38	.042	Unsuccessful	
5	.0222	.28	.039	Unsuccessful	
61	.0281	1.25	.014	Successful	5/8
62	.0281	.78	.022	Unsuccessful	
63	.0247	.72	.019	Unsuccessful	
64	.0219	.58	.019	Successful	
65	.0219	.39	.028	Unsuccessful	

TABLE IV. - STEADY-STATE COMBUSTION PERFORMANCE DATA

(a) Production liner (unmodified)

Run	Simulated rotor speed, percent rated	Simulated altitude, ft	Combustor-inlet static pressure, in. Hg abs	Combustor-inlet temperature, °F	Air flow, lb/sec	Combustor reference velocity, ft/sec	Fuel flow, lb/hr	Fuel-air ratio	Mean combustor-outlet temperature, °F	Combustion efficiency	Nozzle protrusion position, in.
1	70	25,000	28	80	2.7	82	62	0.0064	560	0.93	0
2							115	.0118	915	.97	
3							152	.0156	1105	.91	
4		40,000	15.2	80	1.43	80	47	.0092	710	.92	
5							84	.0164	1070	.82	
6							134	.0261	1440	.75	
7		50,000	9.3	80	.9	82	36	.0111	650	.72	
8							62	.0191	940	.62	
9							77	.0238	1170	.59	
10							102	.0315	1225	.53	
11	70	25,000	28	80	2.7	82	50	.0062	450	.90	5/16
12							97	.010	800	.86	
13							156	.014	1050	.96	
14		40,000	15.2	80	1.43	80	38	.0074	440	.61	
15							67	.0130	920	.88	
16							112	.0218	1250	.78	
17		50,000	9.3	80	.9	82	36	.011	685	.73	
18							58	.018	1000	.72	
19							92	.0284	1340	.65	
20	70	25,000	28	80	2.7	82	49	.005	445	.89	5/8
21							98	.0101	840	1.00	
22							148	.0152	1150	.98	
23		40,000	15.2	80	1.43	80	42	.0082	655	.93	
24							84	.0183	1110	.90	
25							121	.0236	1430	.85	
26		50,000	9.3	80	.9	82	35	.0109	715	.79	
27							53	.0163	950	.75	
28							79	.0244	1160	.65	

(b) Modified liner

Run	Simulated rotor speed, percent rated	Simulated altitude, ft	Combustor-inlet static pressure, in. Hg abs	Combustor-inlet temperature, °F	Air flow, lb/sec	Combustor reference velocity, ft/sec	Fuel flow, lb/hr	Fuel-air ratio	Mean combustor-outlet temperature, °F	Combustion efficiency	Nozzle protrusion position, in.
29	70	25,000	28	80	2.7	82	40	0.0041	365	0.85	0
30							87	.0084	585	.99	
31							90	.0092	755	.99	
32							122	.0126	940	.95	
33							153	.0158	1095	.90	
34							176	.0181	1255	.92	
35		40,000	15.2		1.43	80	30	.0068	435	.80	
36							44	.0086	620	.84	
37							61	.0118	810	.83	
38							76	.0148	960	.82	
39							98	.0186	1160	.82	
40							115	.0223	1315	.79	
41							138	.0268	1420	.73	
42							146	.0284	1450	.70	
43		50,000	9.3		.9	82	31	.0096	490	.57	
44							39	.0116	670	.69	
45							55	.0170	890	.66	
46							78	.0241	1100	.60	
47							95	.0293	1170	.54	
48							99	.0301	1115	.49	
49							103	.0318	1110	.47	

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TABLE IV. - Concluded. STEADY-STATE COMBUSTION PERFORMANCE DATA

(b) Concluded. Modified liner

Run	Simulated rotor speed, percent rated	Simulated altitude, ft	Combustor-inlet static pressure, in. Hg abs	Combustor-inlet temperature, °F	Air flow, lb/sec	Combustor reference velocity, ft/sec	Fuel flow, lb/hr	Fuel-air ratio	Mean combustor-outlet temperature, °F	Combustion efficiency	Nozzle protrusion position, in.
50	70	25,000	28	75	2.7	81	48	0.0050	410	0.85	5/16
51							72	.0074	620	.97	
52							97	.0100	815	.98	
53							127	.0131	1010	.98	
54							161	.0166	1200	.95	
55		40,000	15.2		1.43	79	31	.0060	325	.50	
56							40	.0078	530	.76	
57							52	.0102	735	.88	
58							66	.0128	915	.90	
59							88	.0171	1120	.85	
60							112	.0219	1330	.80	
61							122	.0238	1415	.80	
62							134	.026	1480	.78	
63		50,000	9.3		.9	81	40	.0124	740	.73	
64							31	.0096	560	.67	
65							53	.0164	945	.73	
66							68	.021	1140	.73	
67							76	.0234	1235	.72	
68							84	.0259	1325	.70	
69							100	.031	1400	.62	
70	70	25,000	28	80	2.7	82	39	0.0040	325	0.76	5/8
71							60	.0062	520	.92	
72							71	.0073	620	.99	
73							88	.0091	780	1.00	
74							112	.0116	945	1.01	
75							142	.0146	1125	1.00	
76							159	.0164	1240	1.00	
77		40,000	15.2		1.43	80	35	.0068	510	.83	
78							49	.0095	730	.92	
79							65	.0126	925	.92	
80							104	.0201	1315	.88	
81							120	.0231	1450	.85	
82							135	.0261	1525	.81	
83							27	.0052	340	.62	
84		50,000	9.3	85	.9	83	27	.0083	580	.80	
85							43	.0133	810	.75	
86							64	.0198	1060	.70	
87							83	.0256	1160	.60	
88							91	.0279	1175	.58	

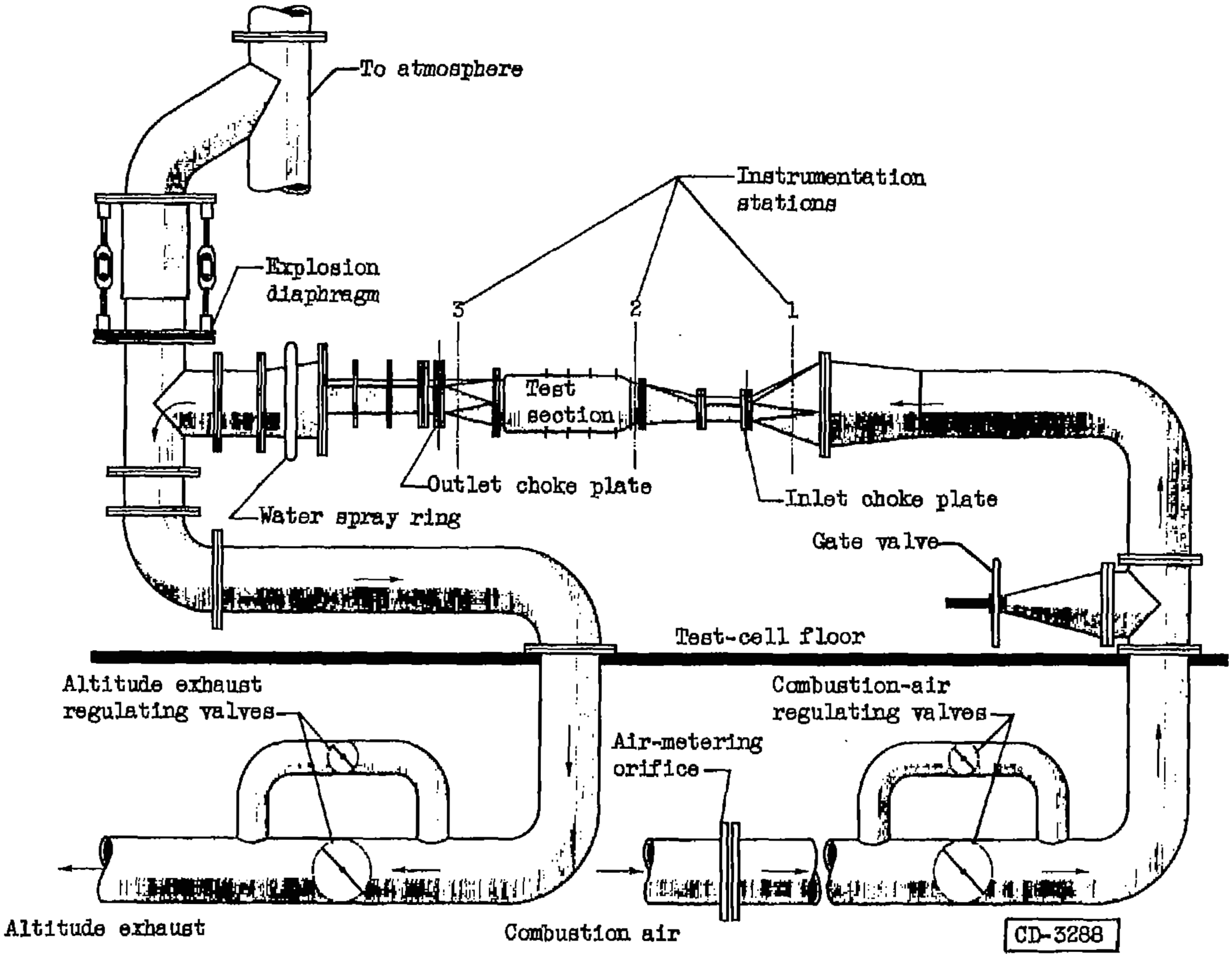

Figure 1. - Diagrammatic sketch of single tubular combustor installation.

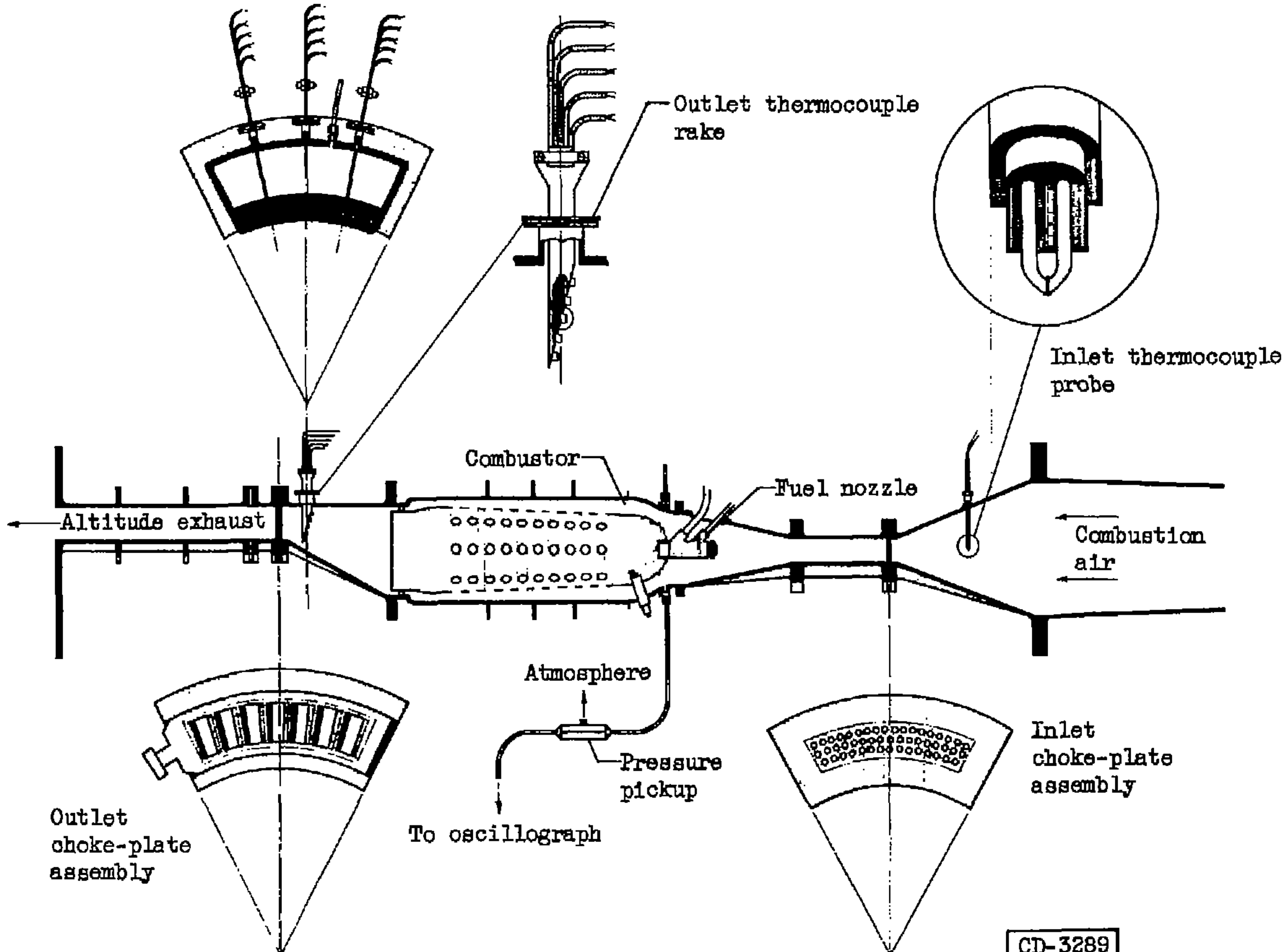

Figure 2. - Instrumentation for acceleration studies.

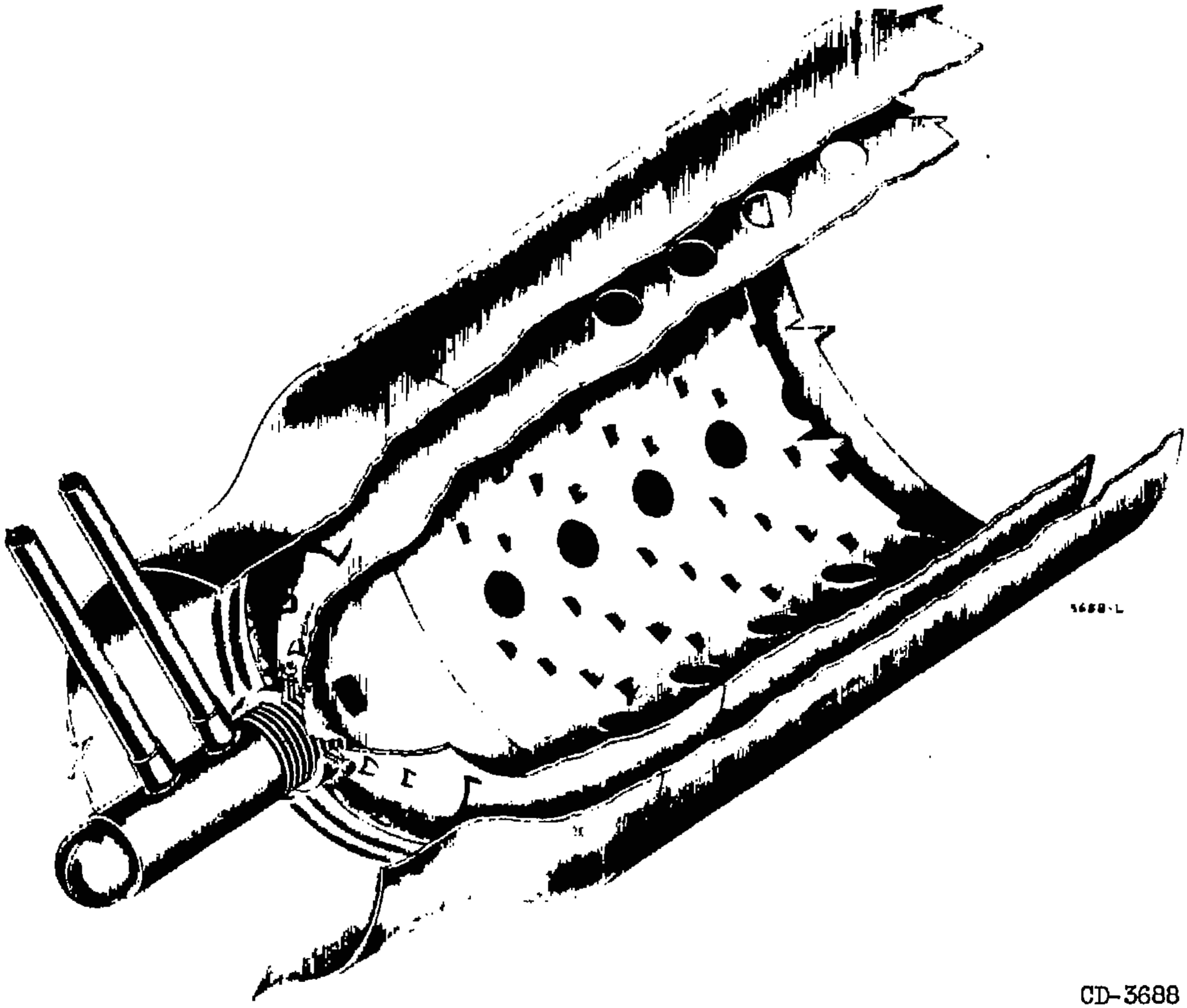

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Figure 3. - Diagrammatic sketch of combustor showing method of attaching liner to nozzle.

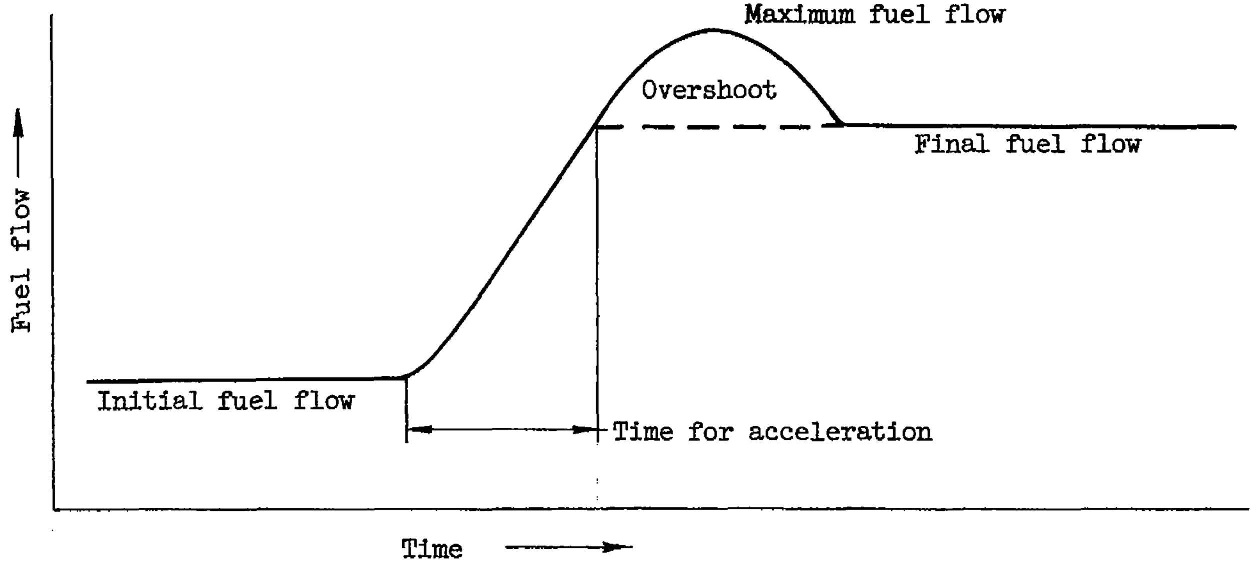

Figure 4. - Typical oscillograph trace of a fuel acceleration (ramp).

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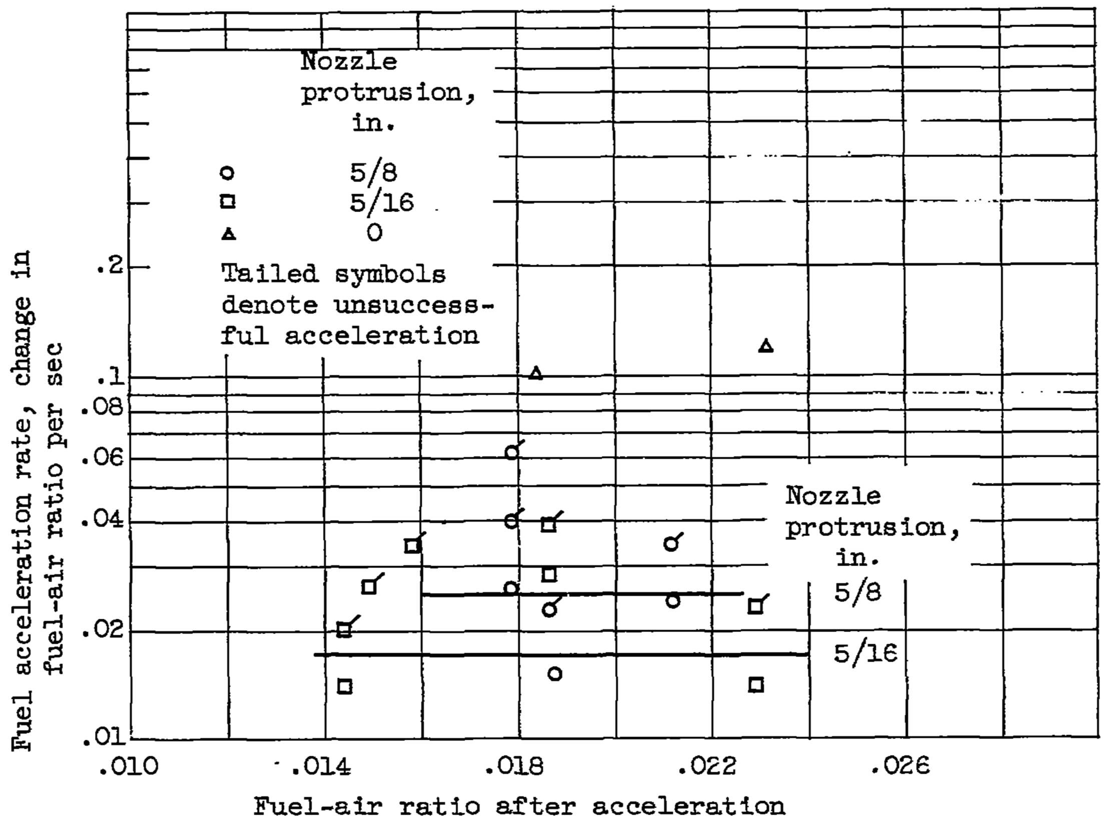
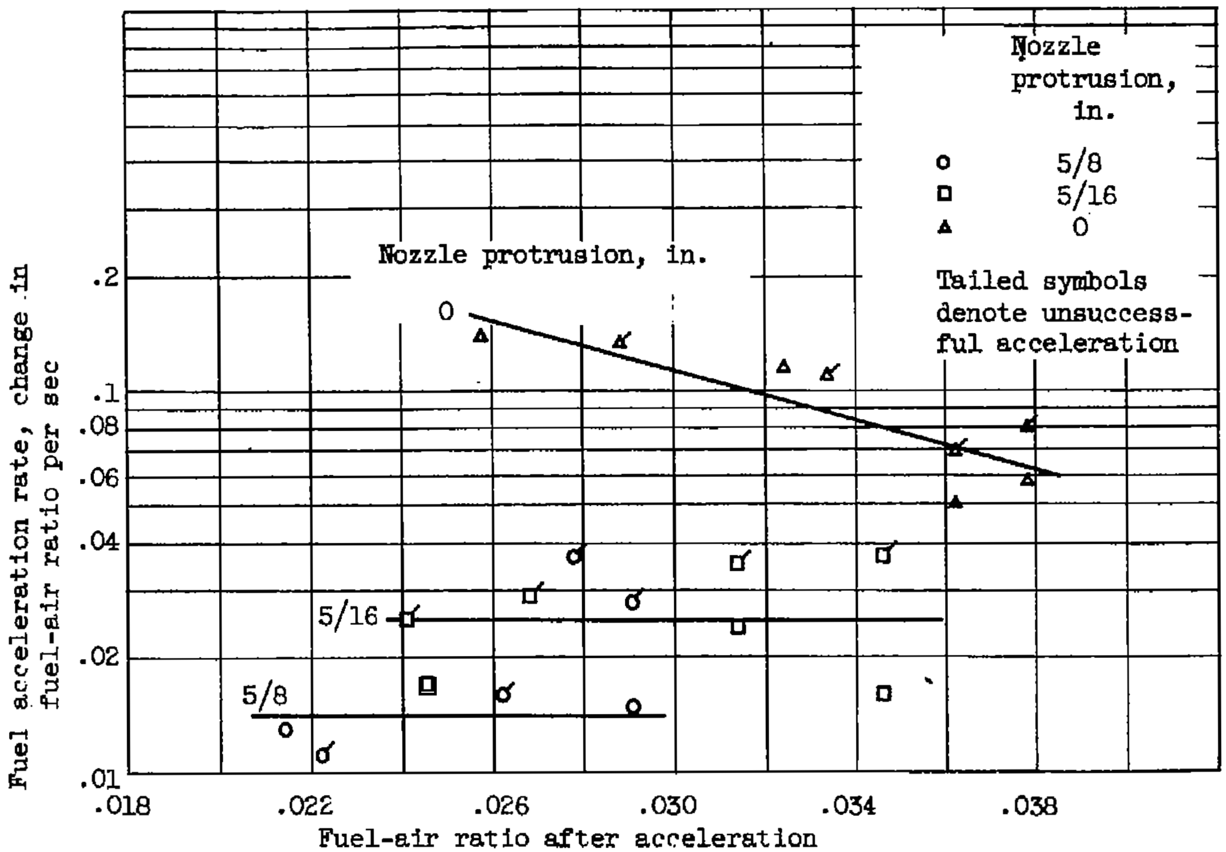
QO-3 back



(a) Altitude, 25,000 feet; approximate initial fuel-air ratio, 0.004.

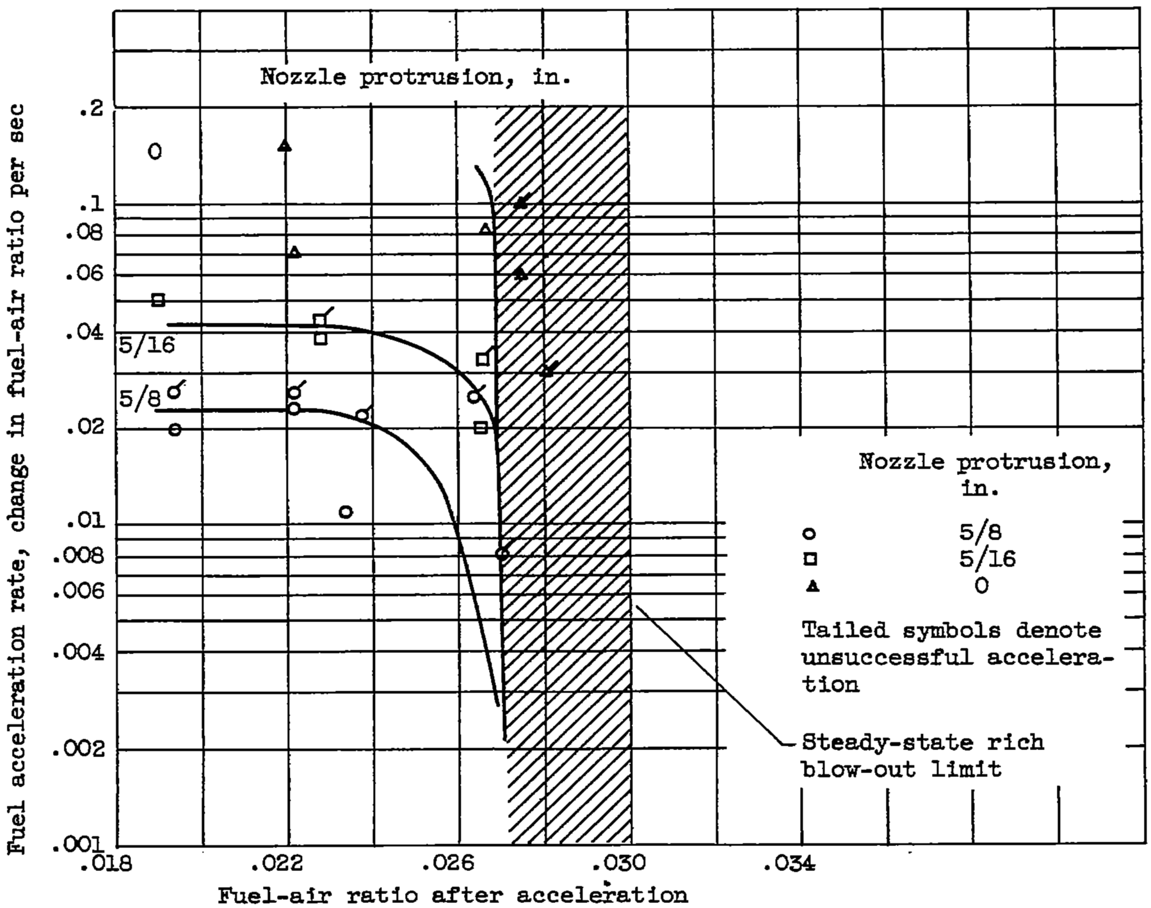
Figure 5. - Combustor fuel acceleration limits for three nozzle protrusion positions at simulated altitude conditions with production liner (unmodified). Rotor speed, 70 percent rated.



(b) Altitude, 40,000 feet; approximate initial fuel-air ratio, 0.009.

Figure 5. - Continued. Combustor fuel acceleration limits for three nozzle protrusion positions at simulated altitude conditions with production liner (unmodified). Rotor speed, 70 percent rated.

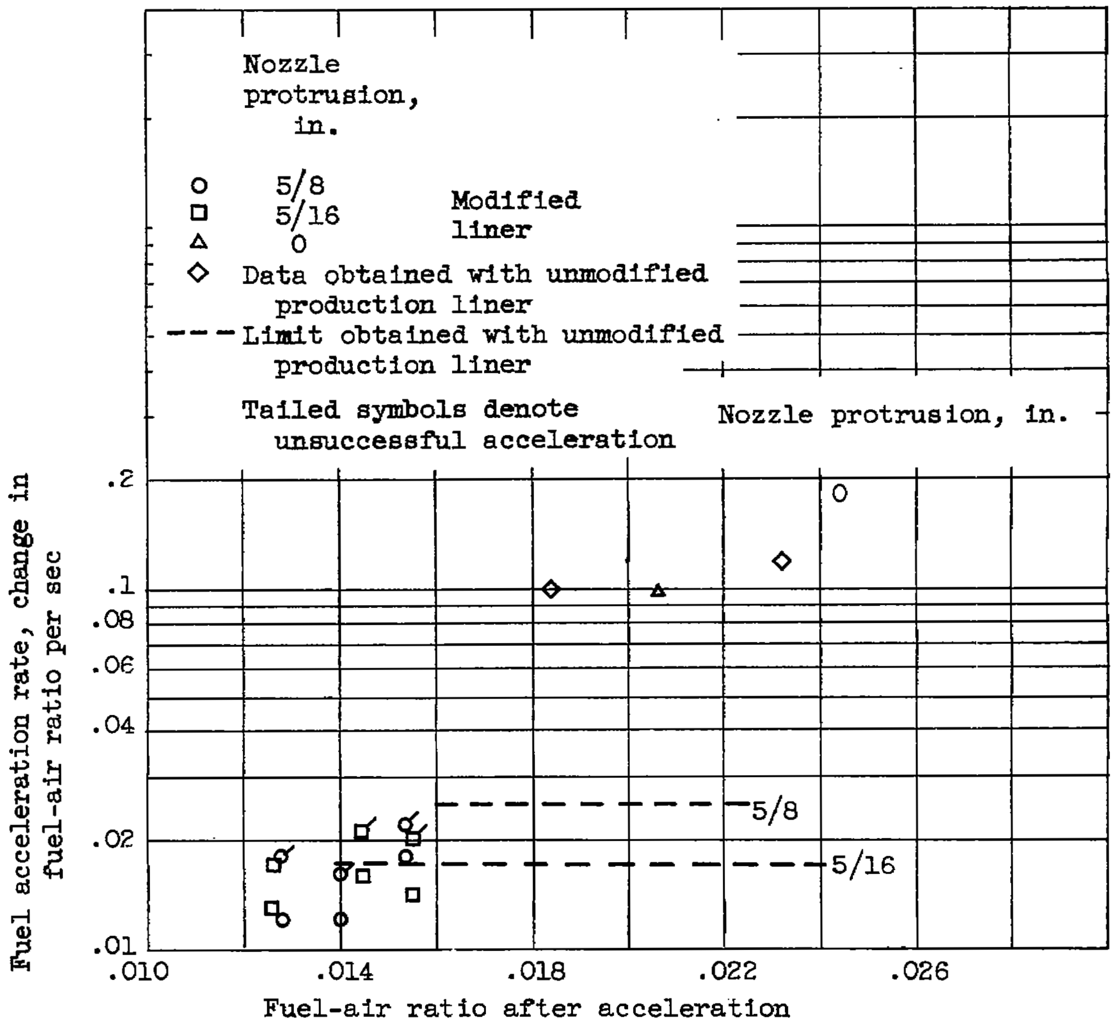
3535



(c) Altitude, 50,000 feet; approximate fuel-air ratio, 0.0115.

Figure 5. - Concluded. Combustor fuel acceleration limits for three nozzle protrusion positions at simulated altitude conditions with production liner (unmodified). Rotor speed, 70 percent rated.

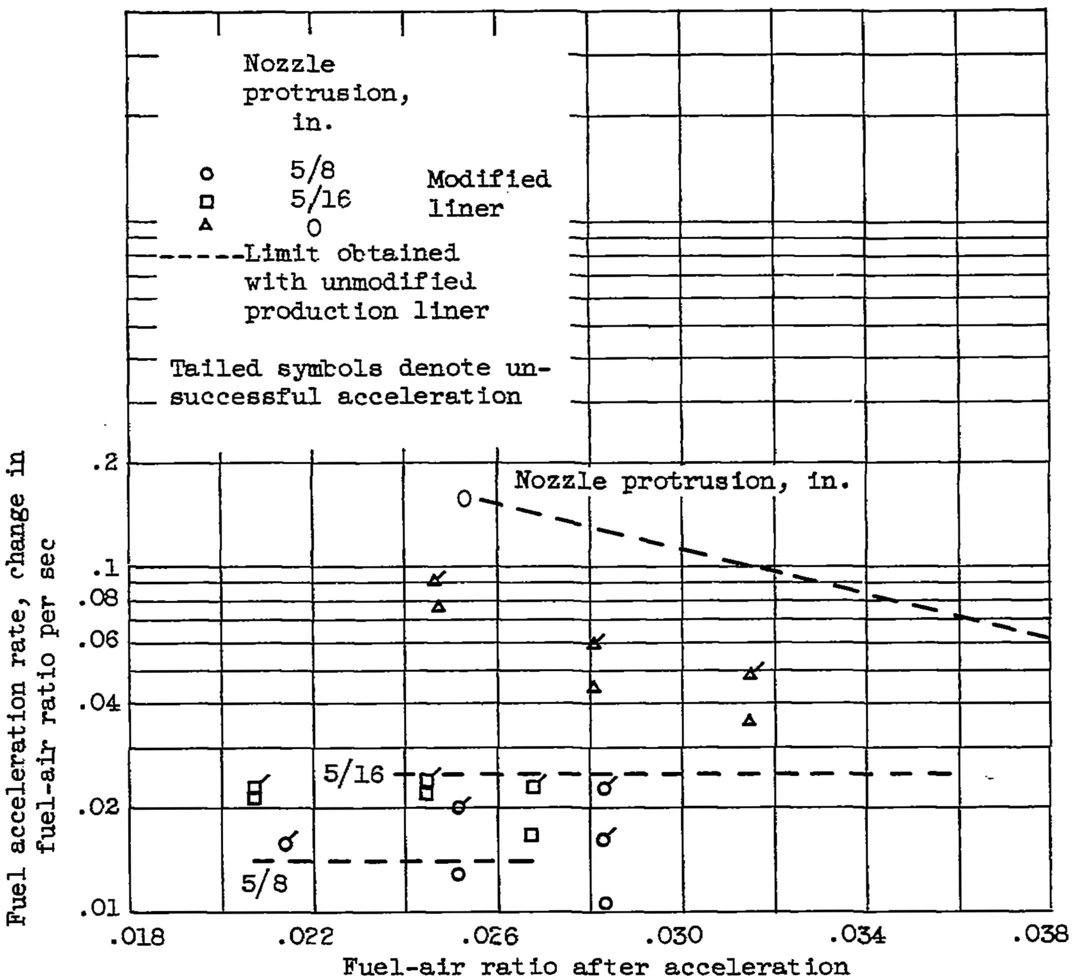
3533



(a) Altitude, 25,000 feet; approximate initial fuel-air ratio, 0.004.

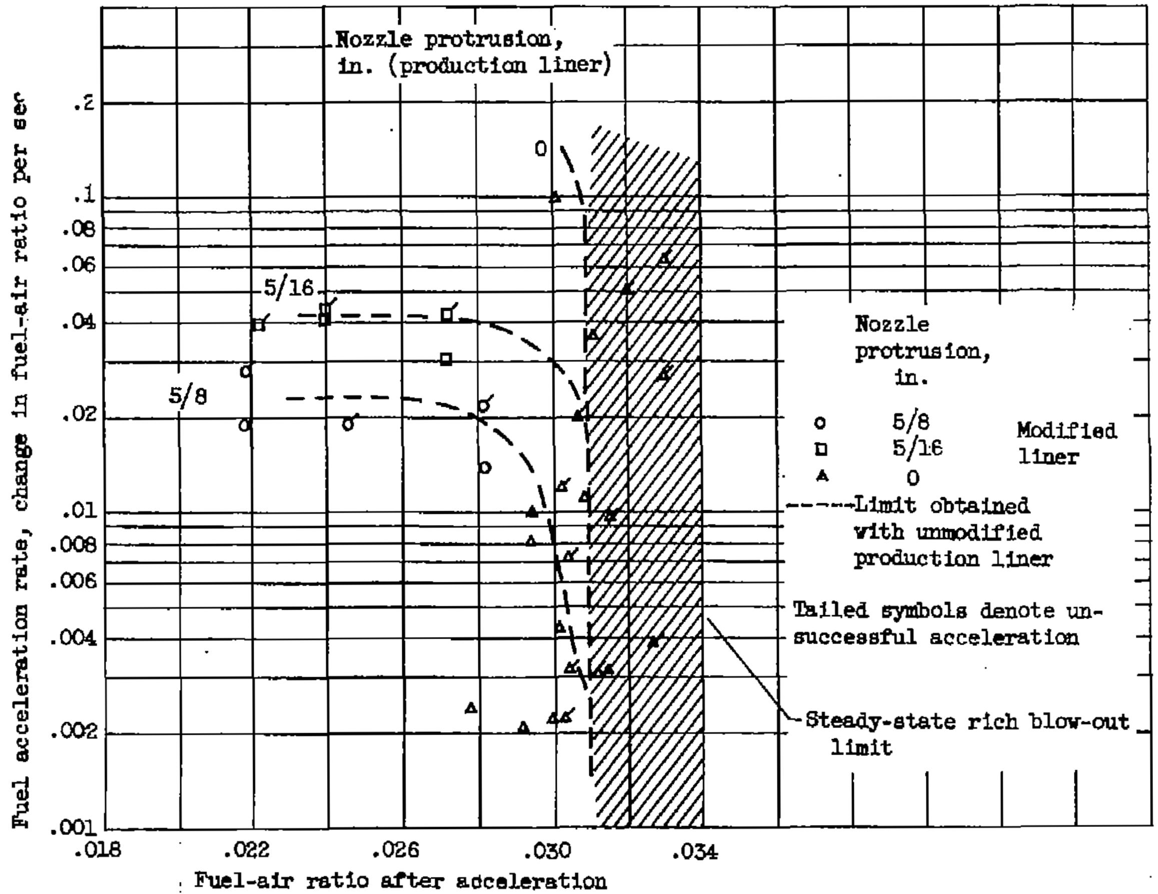
Figure 6. - Comparison of combustor fuel acceleration limits obtained with both unmodified and modified liner for three nozzle protrusion positions at simulated altitude conditions. Rotor speed, 70 percent rated.

3533



(b) Altitude, 40,000 feet; approximate initial fuel-air ratio, 0.009.

Figure 6. - Continued. Comparison of combustor fuel acceleration limits obtained with both unmodified and modified liner for three nozzle protrusion positions at simulated altitude conditions. Rotor speed, 70 percent rated.



(c) Altitude, 50,000 feet; approximate initial fuel-air ratio, 0.0115.

Figure 6. - Concluded. Comparison of combustor fuel acceleration limits obtained with both unmodified and modified liner for three nozzle protrusion positions at simulated altitude conditions. Rotor speed, 70 percent rated..

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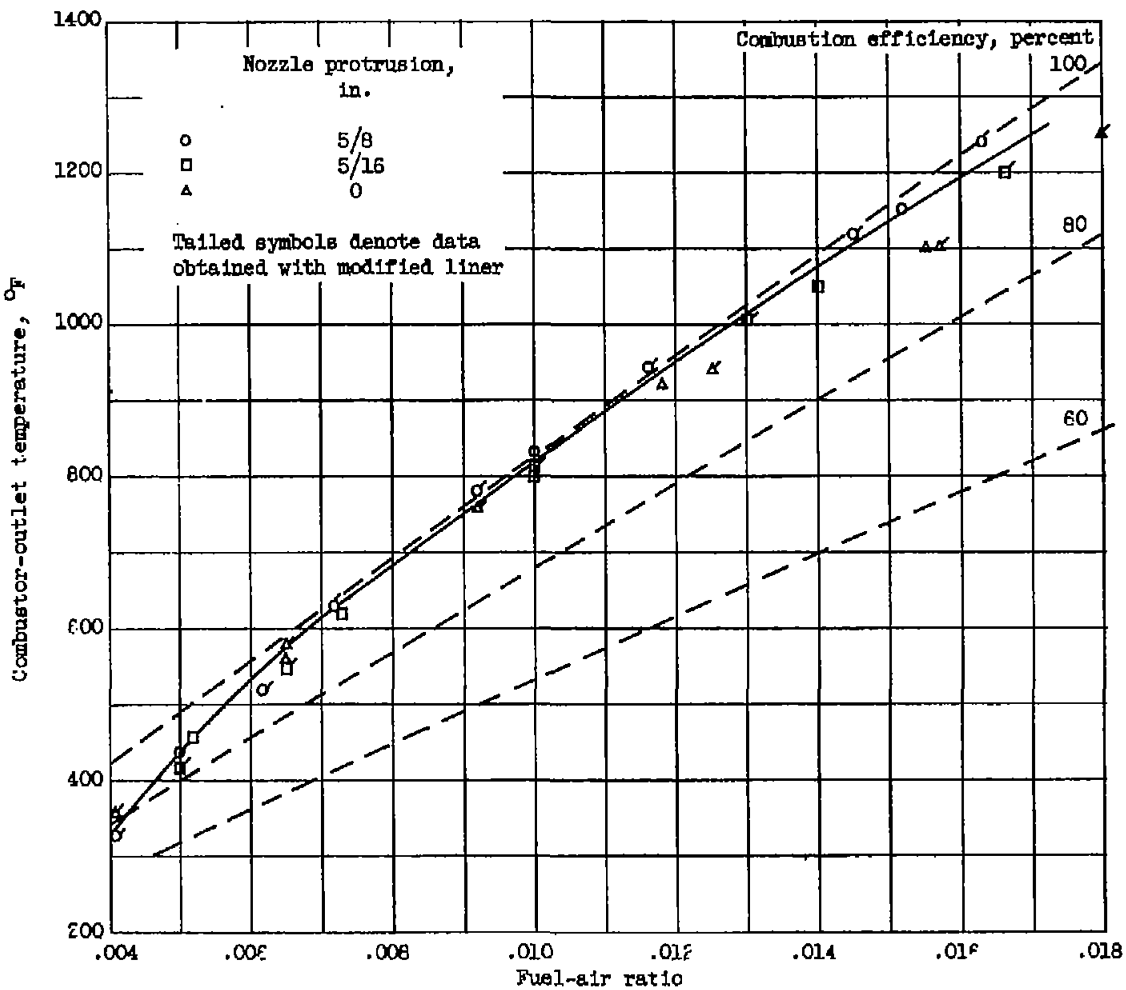

(a) Altitude, 25,000 feet

Figure 7. - Variation of combustor-outlet temperature with fuel-air ratio of tubular combustor operating at simulated altitudes with three nozzle protrusion positions. Rotor speed, 70 percent rated; flight Mach number, 0.

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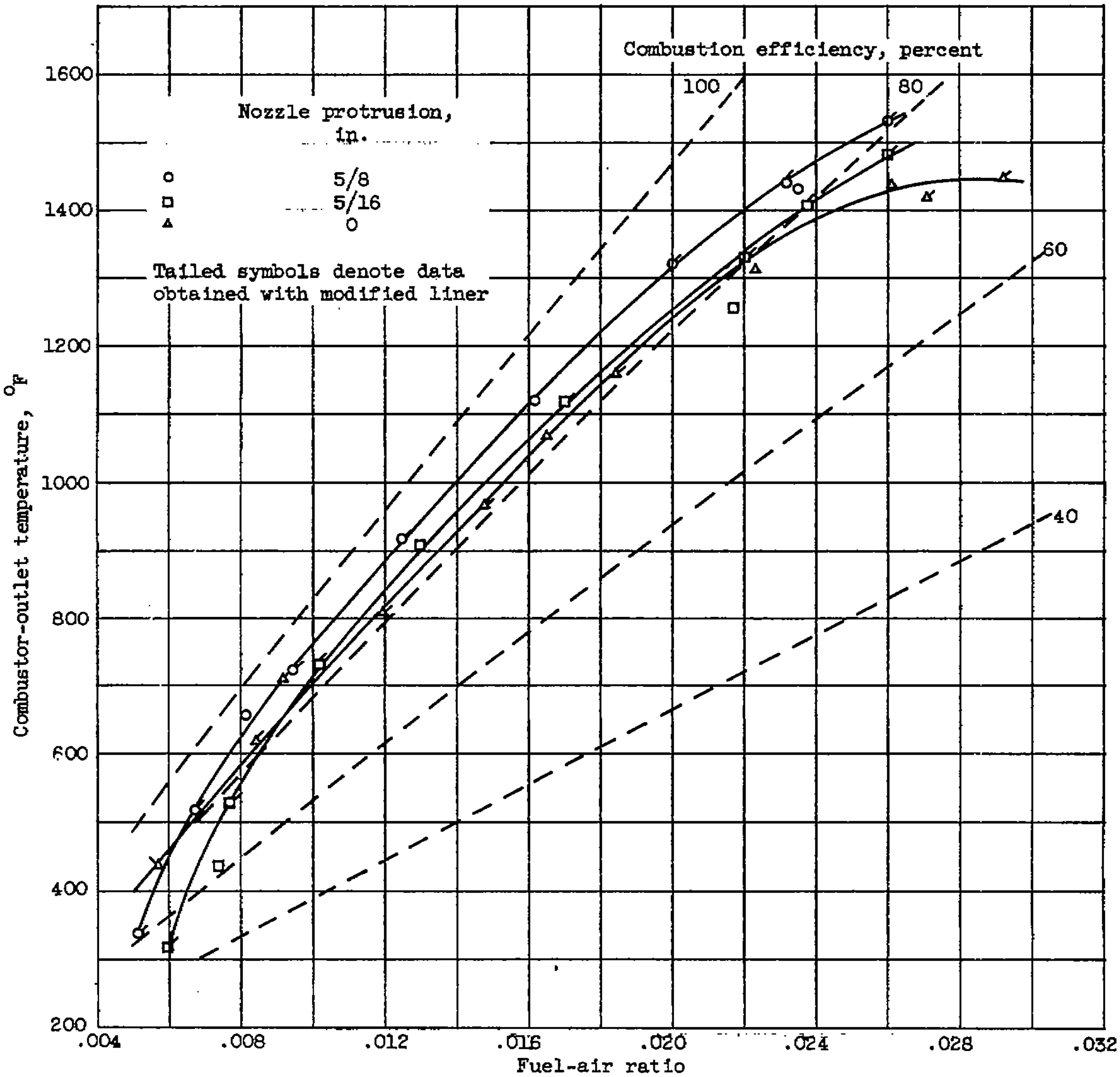
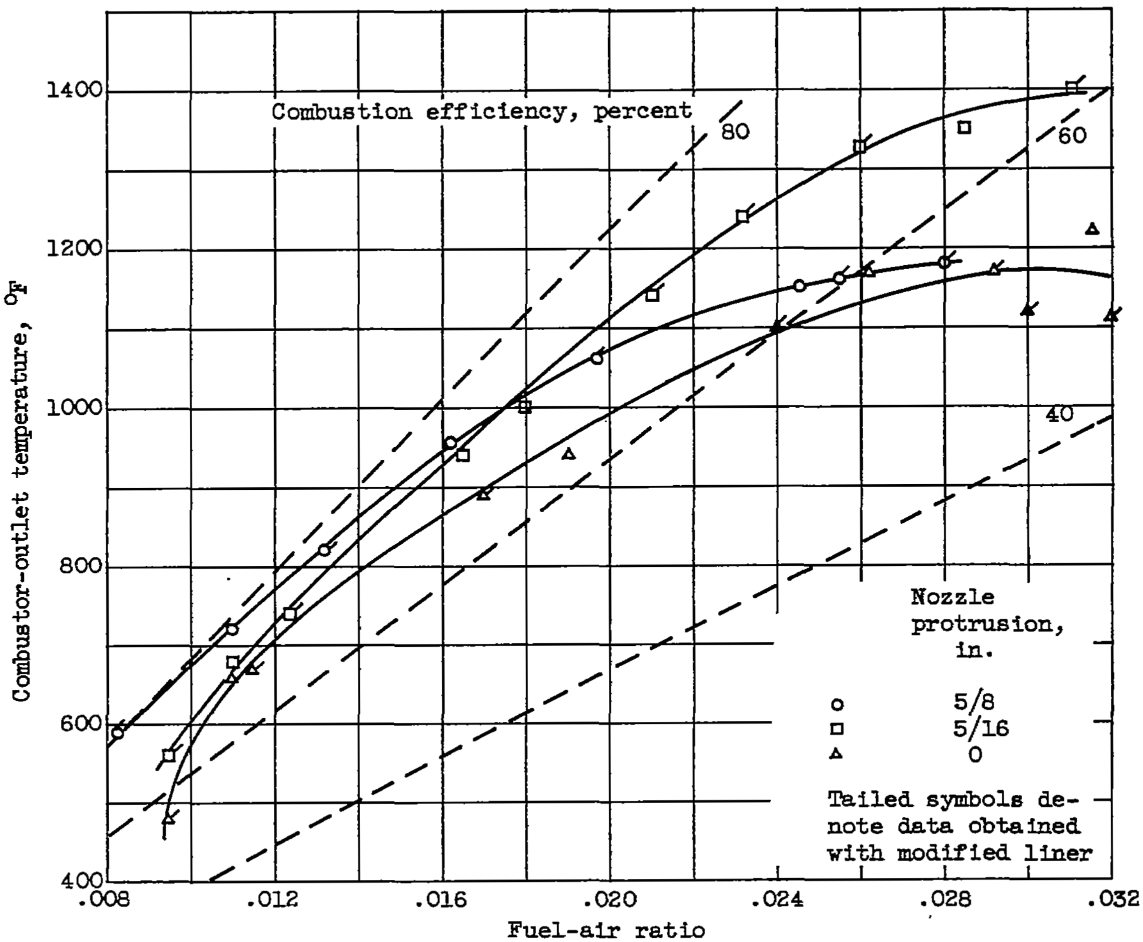


(b) Altitude, 40,000 feet.

Figure 7. - Continued. Variation of combustor-outlet temperature with fuel-air ratio of tubular combustor operating at simulated altitudes with three nozzle protrusion positions. Rotor speed, 70 percent rated; flight Mach number, 0.



(c) Altitude, 50,000 feet.

Figure 7. - Concluded. Variation of combustor-outlet temperature with fuel-air ratio of tubular combustor operating at simulated altitudes with three nozzle protrusion positions. Rotor speed, 70 percent rated; flight Mach number, 0.

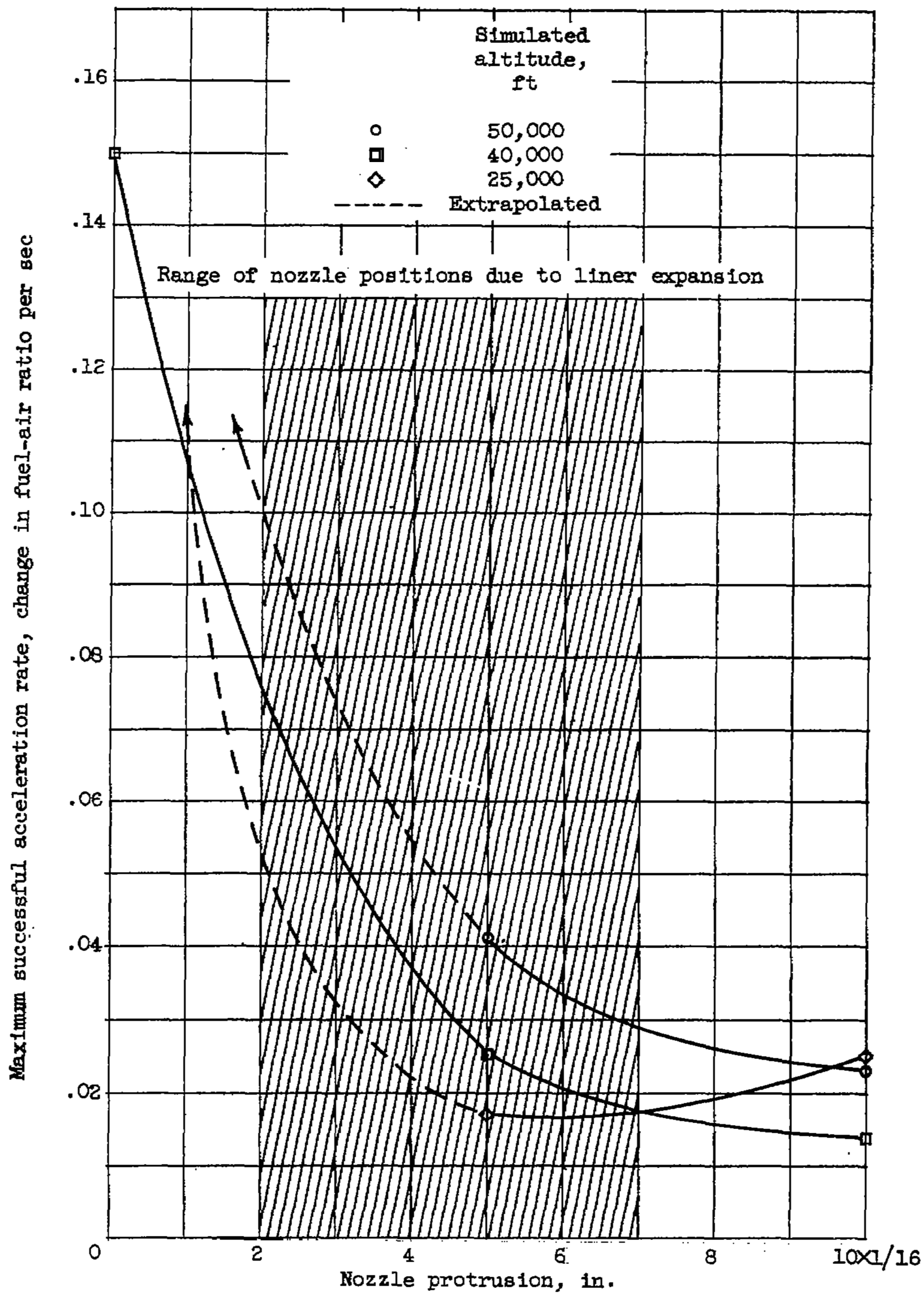


Figure 8. - Comparison of combustor fuel acceleration limits obtained with three nozzle protrusion positions at three simulated altitude conditions. Production liner (unmodified); rotor speed, 70 percent rated; fuel-air ratio after acceleration, 0.026.

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